AGUSTA Experience on Damage Tolerance Evaluation of Helicopter Components

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ABSTRACT

Within the fatigue evaluation of the EH101, Agusta has carried out a specific program of flaw tolerance evaluation of the primary loading path. The program is close to completion and this paper provides a summary of the most relevant results.
For composite components, damage size was increased considering both manufacturing discrepancies greater than the minimum quality standard and impact damages clearly detectable during visual inspections. The favourable data achieved are based on the 'no growth' concept.
The metal parts of the main rotor head were evaluated by enhanced safe life method and fail safe capability. The slow crack growth approach was instead applied for the Rear Fuselage End Fittings, which connect the Tail Unit.
All these evidences can be used in addition to the comprehensive safe life evaluation of the aircraft to improve the maintenance and the repair actions.
Based on this experience, application of flaw tolerance criteria will be carried out on the new helicopters in development phase.

INTRODUCTION

Since 1989 the Civil Rule for Transport Helicopter requires the flaw tolerance evaluation of the fatigue critical parts, wherever practical. For this reason the certification basis of the EH101 was amended with specific requests of Flaw Tolerance evaluation by the three Airworthiness Authorities involved in the program: the Registro Aeronautico Italiano, the British Civil Aviation Authority and the U.S. Federal Aviation Administration. The structural substantiation of the most relevant components of primary loading path was therefore improved to verify their strength in case of improbable and undetected accidental damage.
The reasonable expectance in the near future is that the maintenance manual could take advantage of the flaw tolerance data to improve the evaluation of the inspection intervals by test data and analysis, which properly address the structural strength of damaged parts. The present standard instead takes into account mainly experience and engineering judgement, supported by data from prototypes and lead fleet helicopters. Up to now the purpose to provide a more rational and traceable criteria for the EH101 was achieved by an engineering approach which establishes inspection intervals of the Structural Significant Items (SSI) for accidental damage according to fractions of the retirement life, computed using tables which grade the reduction factor based on the safe life, the static strength margin of safety, the probability of having an accidental damage and the visibility for inspection. Table 1 details the approach for the SSI Accidental Damage Analysis.

An overview of the EH101 flaw tolerance evaluation results is provided hereafter.

MAIN AND TAIL ROTORS

The Main Rotor Head is made by the Main Rotor Hub, with a titanium core and carbon fibre-epoxy loop windings enclosed by glass-epoxy box. Two different configurations of blade grips can be used: the foldable configuration for naval variants, made by Inboard and Outboard Tension Links, with hybrid titanium frames and composite plates, and the 'one-piece' design Civil Tension Link made by a full carbon and glass fibre-epoxy laminate. Most of the Main Rotor Blade shear loads are reacted by the titanium Support Cone, via the Elastomeric Bearings.
The Tail Rotor is a 'see-saw' made by carbon fibre-epoxy Blades and the composite Hub, made by carbon fibre loops windings with glass fibre-epoxy wrap.
Both Main and Tail Rotor have Elastomeric Bearings, providing relevant damage tolerance capabilities.

All the composite parts were evaluated according to AC 20-107A , proving a flaw tolerance demonstration considering:

- Barely Visible Impact Damages (BVID), assumed as the damage visible at the distance of 1 metre caused by dropping a blunt tool of 25 mm diameter, with a maximum realistic energy of 50 Joules, whichever the less;

- environmental ageing consistent with 20 years of service life at T = 24°C and 84% RH

- manufacturing discrepancies at maximum size accepted by production standards, to assure testing components representative of the minimum quality standard.

Static strength was proved at Ultimate Load at the end of the service life.
Additional evaluations for damage tolerance were carried out increasing damage size to a level clearly detectable by visual inspections or exceeding the size of the manufacturing discrepancies beyond the production standards, with the purpose to improve the Accidental Damage Analysis.
Metal parts were evaluated for fatigue according to safe life approach. Additional evaluations were carried out to address the flaw tolerance capabilities for Clearly Detectable Damage of the M.R. Hub titanium core, the Damper Hub Attachments, the Support Cones and the Tension Links titanium frames.

The Table 2 summarises the Agusta components of the EH101 which were evaluated for Damage Tolerance.
Figure 1 points out the Main Rotor Head components evaluated for Damage Tolerance capabilities.

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M.R. Hub

The M.R. Hub, figure 2, assures connection of the 5 blades and transfers the loads to the mast through a splined coupling. It is a hybrid metal-composite structure made by a titanium core and 10 small plus 2 large carbon fibre-epoxy loop windings enclosed by an external wrapped glass-epoxy box. The hub is a 125°C autoclave cured component.

Composite and metallic parts were evaluated for DT capabilities according to a full scale spectrum test with intentionally induced manufacturing discrepancies and localised fatigue failures due to the safe life testing phase.
No damage growth in the service life was proved by tests in composite parts, which were impacted for Barely Visible Impact Damage (BVID) at energy of 50 J with a spherical impactor ϕ=25 mm, and for Clearly Visible Impact Damages (CVID) with a sharp tool made by a 90° pyramid, figure 3.
In addition, scratches and delaminations in the glass wrap due to amplified loading of safe life test showed a slow flaw growth behaviour in the start-stop cycles of the D.T. testing phase, and no damage growth at flight loads, figure 4. The test achieved the target of 10.000 hours for the Navy spectrum, the most severe for the higher start stop loads due to the blade folding cycle, proving a safe inspection interval of 2500 hours using a life safety factor of 4.

After the safe life test, tolerance to fretting and fail safe capabilities in the titanium core were checked removing the solid lubricant and 10% of the Hub teeth in the splined coupling with the M.R. Mast and testing at amplified Torque, figure 5. The test achieved the Civil variant design target of 40.000 h for the start-stop torque loading cycles, proving also no high frequency fatigue damage by the S/N curve for the fretting failure mode. A mean strength reduction by 2 standard deviations, appropriate for Enhanced Safe Life approach, was used instead of the normal reduction by 3 standard deviations for pristine components, considering the structural redundancy demonstrated and the relevant flaws under test.
Flaw Tolerance capabilities of the Aluminium Upper and Lower Plates was checked removing 4 out of 15 bolts on both plates in the splined area and testing the Support Cone coupling area with relevant cracks and fretting due to the initial safe life phase of testing, figure 6. The Al Plates proved fail safe capabilities for bolts failures achieving the target for both Civil and Navy usage spectra and proved in addition more than 1000 hours of Enhanced Safe Life for the Support Cone coupling area with severe fretting and cracks. The pristine component has unlimited fatigue life and this evaluation could be used to improve the evaluation of inspection intervals in service for Clearly Detectable Damages.

In addition, the M.R. Hub fatigue test proved the capability of the composite arm to sustain for at least 3 flight hours the full spectrum of shear loads of the M.R. Blade. This should allow, from the structural point, the safe landing of the aircraft in the remote event of a Support Cone failure, considered immediately detectable in flight due to the increased vibration level. Residual strength up to limit load was proved without failure of the composite arm, showing also that no stiffness degradation occurred in the M.R. Hub composite parts.

M.R. Damper Hub Attachments

The M.R. Damper Hub Attachment is made by an upper and a lower Al-Alloy plate, bonded to the Hub and additionally fitted by three bolts, figure 7.
After having initiated a crack in the safe life test at amplified loads, the crack grew in the upper plate reaching the bolt hole. No further crack initiation occurred for 200 flight hours at max flight load, using $1/(1-2\sigma)$ load safety factor, appropriate for the Enhanced Safe Life evaluation. Residual strength was proved up to limit load, considering the partial failure already occurred.
An additional Rogue Flaw was intentionally made by a corner saw cut of 2 mm depth in the lower plate, in the section which has showed a further failure mode during the safe life tests. Even in this case the 200 h inspection interval was confirmed. The evaluation was not improved since both failure modes are easily detectable during visual inspections of the M.R. Hub.

M.R. Support Cone

The M.R. Support Cone connects the Pitch Change Arm to the Rotor Hub. It is made by Titanium with a vulcanised Elastomeric Centering Bearing, which allows the pitch movement of the blade, figure 8. The Support Cone is connected to the Hub core via two Special Bolts and four Stud Bolts.
The structural significant sections were taken into account for the flaw tolerance evaluation of this component, according to the failure modes showed in the safe life tests:

- the Stud Bolts failure, which was proved as not fatigue critical due to redundancy of the two Special Bolts
- fretting of the Special Bolts
- fretting in the lugs of the Special Bolts
- flaws in the lugs of the Special Bolts, simulating flaws that could be done during assembling and maintenance operations, figure 9. Flaws made by 'V' grooves of 0.35 and rogue flaw 0.50 mm dept were made in the test components using a sharp tool to minimise plastic deformation at the flaw tip. Constant amplitude loading tests were carried out to determine the Wohler curve of the flawed specimens for the 'Enhanced Safe Life' calculations.
- the same flaws were made also in the cylindrical part of the Support Cone, for simulation of improper manufacturing or maintenance operations.

The retirement life of the component is determined by the fretting failure mode in the Support Cone lug.
The rogue flaw in the lug reduces the fatigue strength of the pristine components by 1.25, the flaws in the other sections did not show relevant fatigue strength reduction. The Enhanced Safe Life evaluation for surface flaws in the lug did not show any additional fatigue life reduction. Figure 10 shows the fatigue test data.

M.R. Inboard and Outboard Tension Links

The M.R. Inboard Tension Link is a hybrid structure composed by two carbon and glass-epoxy composite plates bonded and bolted to a titanium forged frame in the outer folding section.
The M.R. Outboard Tension Link is a hybrid structure composed by a titanium forged frame with two glass-epoxy composite plates bonded to the upper and lower metal frame plates.
Figure 11 shows the two components, designed to allow blade folding for the Navy requirements.
Flaw Tolerance of the composite plates to 50 J impact damages, manufacturing discrepancies and environmental ageing were proved to address the retirement life, proving also ultimate load static strength after fatigue in hot-wet ageing.
Tolerance to CVID by sharp impact at 50 J was additionally proved by full size test for 500 flight hours with residual static

strength test up to limit load. Additional tests of composite lug structural elements have not shown fatigue strength reduction between test specimens with impact damages and with manufacturing discrepancies greater than production quality standards.
Full size lug structural elements were tested for evaluation of fatigue strength reduction due to rogue flaws in the lugs of the folding section. The specimens were flawed by 'V' grooves 0.5 mm deep, made by milling machine, simulating severe scratches in the lug contact sections. The scratches did not reduce the fatigue strength if they were made out of the net-tension area. An additional type of flaw was evaluated considering improper repair, simulated in the lug elements by 'U' grooves made by grinding wheel. The test data proved a relevant fatigue life reduction, but without additional reduction of the endurance value. Figure 12 compares the Wohler curves.
Full size tests of the Outboard Tension Link verified the capability of the composite plate to provide a redundant loading path after local failure of the titanium lug, figure 13, proving at least 40 hours of fail safe capabilities at maximum flight loads and residual static strength up to limit loads. An Equivalent capability is expected by the outer lug section of the Inboard Tension Link.
In both components the fatigue crack in the titanium lug is detectable by detailed inspection.

M.R. Civil Tension Link

The M.R. Civil Tension Link is a hybrid structure made by carbon and glass-epoxy composite plates, very similar to the Inboard Tension Link used on Navy configuration. This is a full composite component Civil Certified according to the Flaw Tolerance evaluation taking into account 50 J impact damages, manufacturing discrepancies and environmental ageing and proving ultimate load static strength after fatigue in hot-wet ageing.
Tolerance to CVID by sharp impact at 50 J was additionally proved by full size test for 500 flight hours with residual static strength test up to limit load, figure14.

Elastomeric Bearings

Two types of Elastomeric Bearings, Centering and Spherical, are installed on the M.R. Head, providing the drag, pitch and flap hinges of articulation and transmitting the blade shear loads to the Hub. The Spherical Bearing transmits part of the blade shear loads due to its radial stiffness and it can provide a redundant loading path for the flight completion in case of failure of the Centering Bearing, as proved by test.
The Tail Rotor Elastomeric Thrust Bearing allows blade motions and the coupling with the T.R. Hub.
The qualification tests provided data for the durability assessment and relevant evidences supporting the slow damage growth capability taken into account to address the periodic inspections. The driving parameter is the failure mode shown during tests with a progressive wearing out of the elastomer, with rubber loss.

An additional test was carried out to provide comprehensive data on the strength and stiffness degradation after fatigue damage, high temperature and contamination by potentially aggressive fluids, like hydraulic fluid and lubricating oil. The tests followed three phases: accelerated fatigue damage at amplified loads, damage propagation monitoring with and without contamination, residual static test for strength and stiffness evaluation up to limit load, figure 15. Initially two specimens were planned, but due to the very positive results all the test plan could be carried out with one specimen only, covering the routine inspection intervals for fluid contamination and improving to 900 flight hours the evaluation of slow structural degradation due to fatigue loading, initially set at 300 h. A life safety factor of 3 was considered due to structural redundancy.
The Tail Rotor Bearing was used since this provides the higher ratio between external surface exposed to contamination and rubber volume.

Tail Rotor Hub

The Tail Rotor Hub is made by a carbon fibre-epoxy loop windings with internal glass-epoxy plates, wrapped by a glass-epoxy cross ply. The Flaw Tolerance to 50 J impact damages, figure 16, manufacturing discrepancies and environmental ageing was proved by tests to address the retirement life, proving also ultimate load static strength after fatigue in hot-wet ageing.
A critical sections are protected in service by the blade cuff and therefore no additional investigation for Clearly Visible Damages was carried out.

Tail Rotor Blade

The Tail Rotor Blade is made by carbon fibre–epoxy. In addition to the Flaw Tolerance evaluation of the Spar, considering BVID up to 50 J, manufacturing discrepancies and environmental ageing, a specific test was carried out to address the fail safe capability for large cracks in the trailing edge skin, covering 10 flight hours of spectrum test factored for scatter and environment with the trailing edge removed from the specimen, figure 17. Residual static strength was then proved up to limit load.
A daily visual inspection for cracks in the trailing edge was required, without any special tool, covering the accidental damage on the blade skin.

Rear Fuselage

Three different variants of Rear Fuselage were designed according to three basic configuration of the aircraft: Civil, Utility and Naval. The structures are conventional metal fuselages made by Al-Alloys stringers and skin panels. The Utility variant has a ramp door for cargo operations.
Fail safe capabilities for the airframe were preliminary evaluated supporting 1000 hours of visual inspection and will be improved by a comprehensive analysis.

The most relevant evidences for damage tolerance were provided by the spectrum test of the Naval subcomponent, made by the Rear Folding Beams and the last bays of the fuselage. This configuration provides a foldable joint with the Tail Unit structure. In the safe life test a rig failure occurred during the folding phase, overloading the Rear Upper Folding Beam which failed with a large crack which cut through the Beam critical section, near the Actuator Lug, figure 18. The test was continued for the flight loading phase proving 50 hours of fail safe capability, with a life safety factor of 4. The crack is easily detectable by visual inspection while the Tail Unit is folded and this has an average occurrence of 4 hours in the usage spectrum.

The equivalent subcomponent for the Civil variant showed 500 hours of slow crack growth for fatigue crack starting from the fillet of the boss in the development configuration, figure 19. Inspection by Ultrasonic can detect the crack with the Tail Unit installed.

Tail Unit Structure

The composite Tail Unit is a structure made by panels of carbon-epoxy skins with honeycomb core. Metal attachments are used in the coupling areas with the gearboxes, the tailplane and the forward structure (the Rear Fuselage). The skins are jointed using both bonding and rivets, to provide redundant load path in case of weak bonding. The Tail Unit was substantiated by analysis and tests, using test data on structural elements, subcomponents and full size specimens representative of the structure in dry and environmentally aged conditions.
Tolerance to Barely Visible Impact Damages (BVID) was proved by static and fatigue tests up to conservative but still realistic energies established by an hazard assessment, with a maximum energy level of 50 J near metal attachments and a minimum of 10 J on upper sandwich panel, protected in service by the fairings.
Tolerance to manufacturing discrepancies was proved by test using items properly manufactured with artificial flaws, like Teflon inserts for delaminations and debondings.
Tolerance to moisture absorption due to hot-wet environment was demonstrated by static and spectrum tests of subcomponents after accelerated aging up to moisture saturation at 84%RH, since the low thickness skin could reach moisture saturation in service. This was determined by diffusivity calculations according to Fick's law at the worst hygrothermal environment assumed of 26°C of temperature and 84% RH.

After local failure caused by safe life tests, subcomponents were fatigue tested to evaluate their tolerance to delaminations, disbondings, rivet failures and cracks in the skins or in the metal attachments caused by amplified loading or intentionally done after the basic fatigue tests.

These tests were related to the vertical fin and the tailplane attachments area, figures 20 and 21. They proved no flaw growth or redundancy of the loading path in sections of load concentration.

Additional tests for the composite structure were carried out on damaged panels and structural elements as a relevant background for selection of artificial damages to be applied in the subcomponent and full size test. The comprehensive evaluation was carried out by the spectrum tests on the full size structure, after completion of the one lifetime safe life test with BVID and manufacturing discrepancies. Sharp impacts were added to the structure at the maximum energy established by the hazard assessment, causing Clearly Detectable Damages, mainly skin indentation and core crushing. Considering the low probability of having the combination of a highly damaged structure with bottom of scatter strength, the DT test was carried out with safety factors computed on a B-basis. An additional load amplification factor of 1.1 was applied to account for environmental conditioning. Residual static strength was checked at limit load.

Since the loading spectrum has relevant occurrences of both maneuver and vibratory loads, no practical application of slow flaw growth capability was envisaged, since damage would grow very fast, and the test was therefore carried out to check the no growth or the structural redundancy.
An inspection interval of 600 flight hours was proved for Clearly Detectable Damages.

CONCLUDING REMARKS

The helicopter loading spectra of the dynamic components are strongly affected by rotor induced loads, which have relevant amplitude and high frequency. If damage grows due to the flight loads from a size detectable in service, the crack propagation rate would require frequent inspections in single load path components.

For this reason most of the evaluations carried out for the EH101 main load path were focused on 'no damage growth' approach. This was determined for composite parts mainly by testing of damaged components, addressing the retirement life with BVID, environmental aging effects and manufacturing discrepancies covering the minimum quality item. The flaw tolerant evaluation was then improved by increasing the damage size considering Clearly Detectable Damages, and verifying an applicable inspection interval by no growth demonstrations or by proving structural redundancy.

The Enhanced Safe Life methodology was applied for metal parts to establish a time to crack initiation from relevant flaws, clearly detectable by inspection.

Based on this experience Agusta is addressing the flaw tolerance evaluation for NH90 and the substantiation criteria of the new helicopter AB139.

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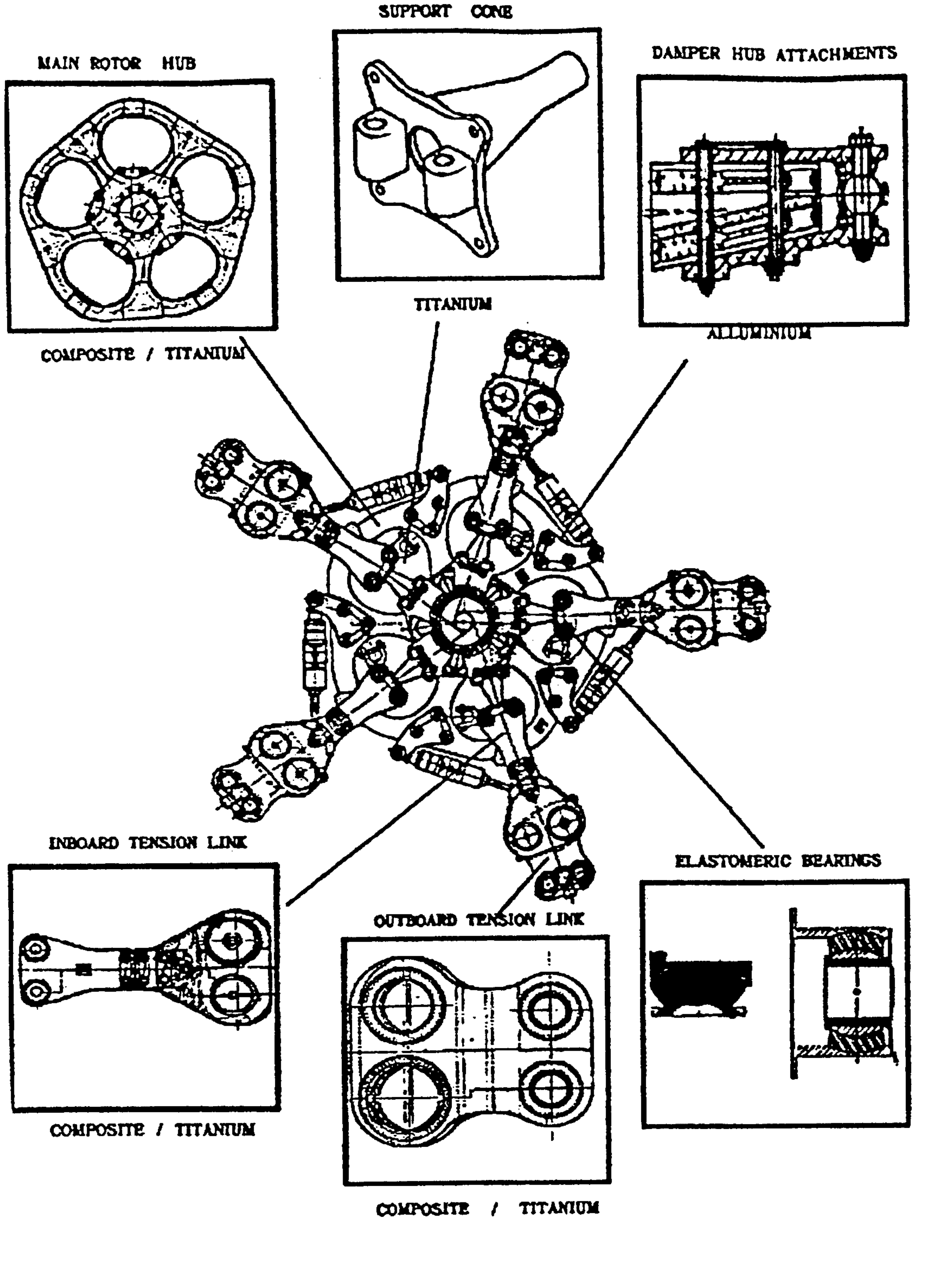

Figure 1 - EH101 Main Rotor Head - Components verified for Damage Tolerance capabilities.

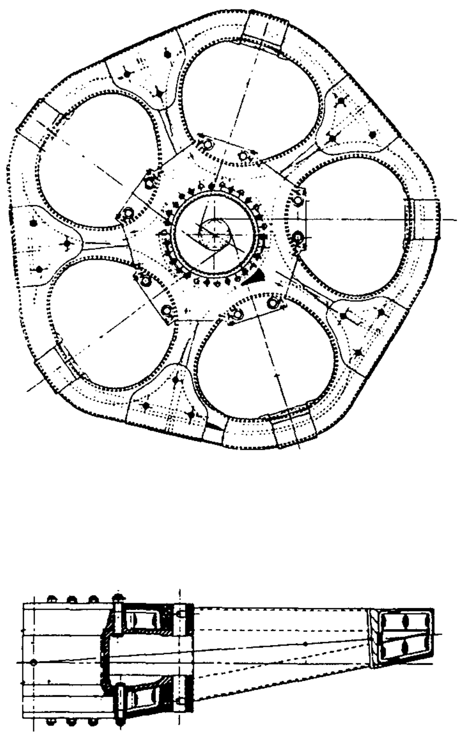

Figure 2 - Main Rotor Hub

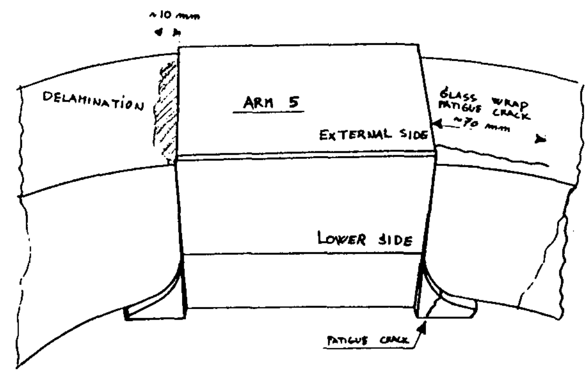

Figure 4 - M.R. Hub Damages after Safe Life Test

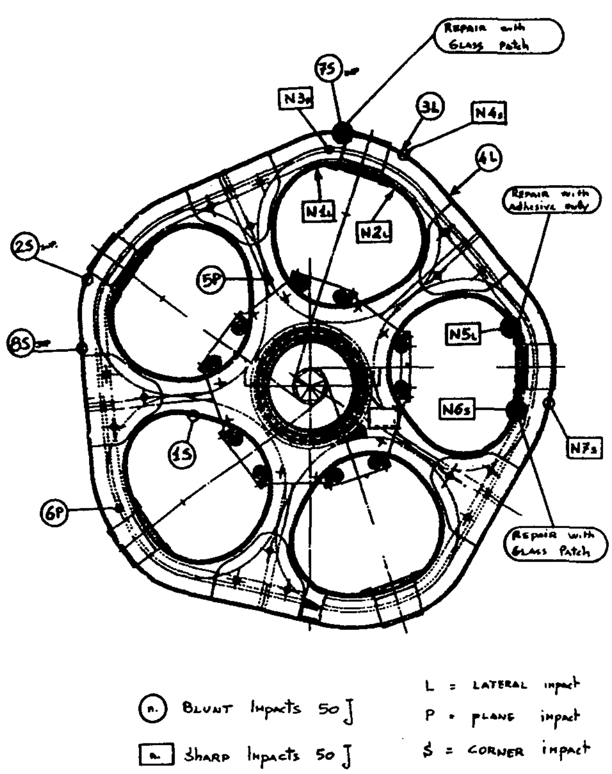

Figure 3 - M.R. Hub Impact Damages

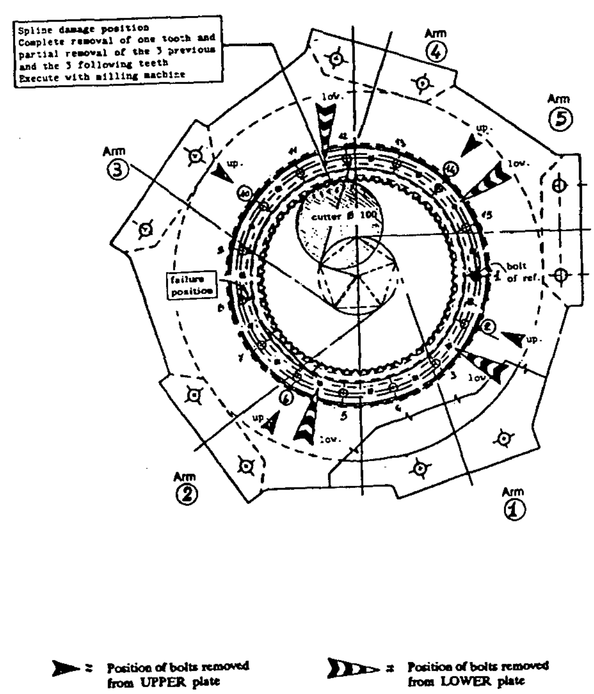

Figure 5 – Flaws in M.R. Hub Splined Coupling

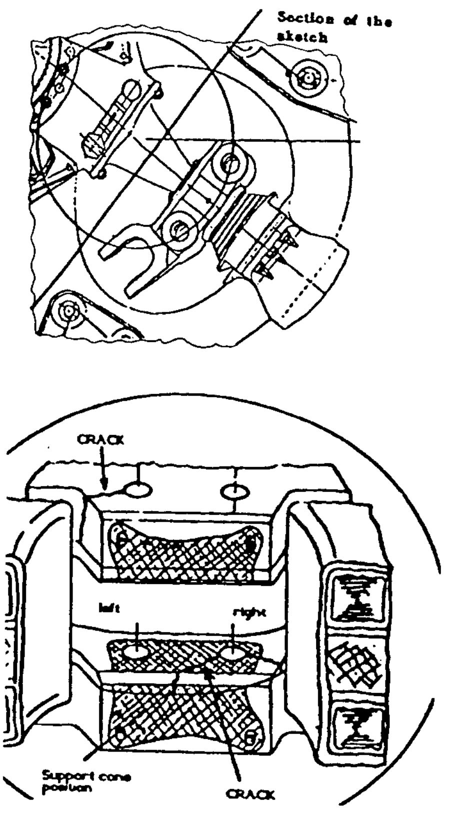

Figure 6 - M.R. Hub Flaws in Aluminium Plates and Titanium Core

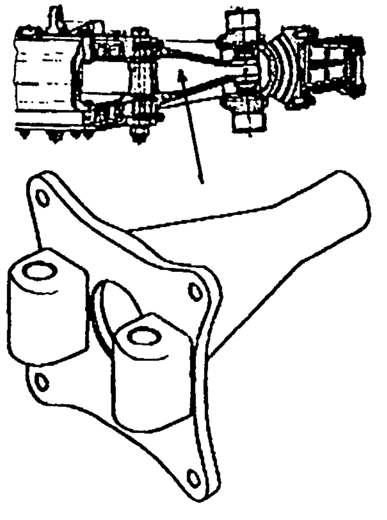

Figure 8 - M.R. Support Cone

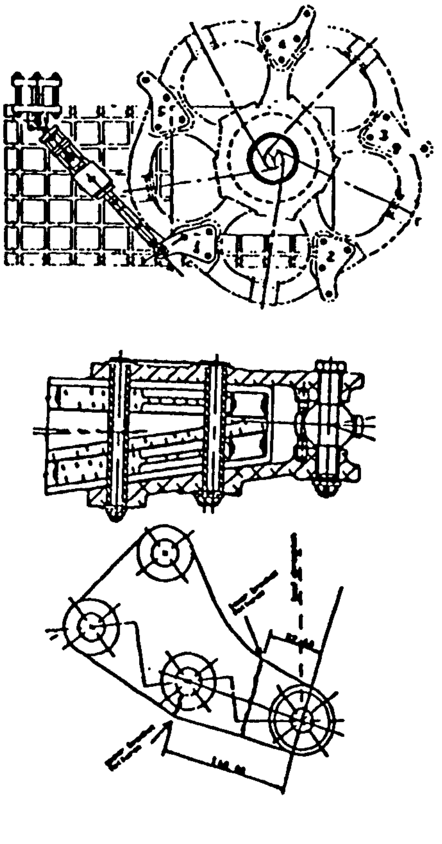

Figure 7 - M.R. Damper Hub Attachments

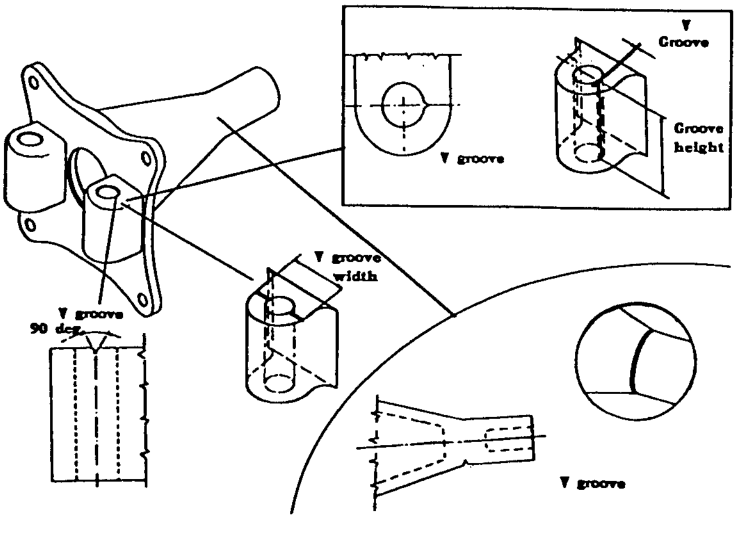

Figure 9 - M.R. Hub Support Cone - 'V' grooves of flawed components for fatigue tests

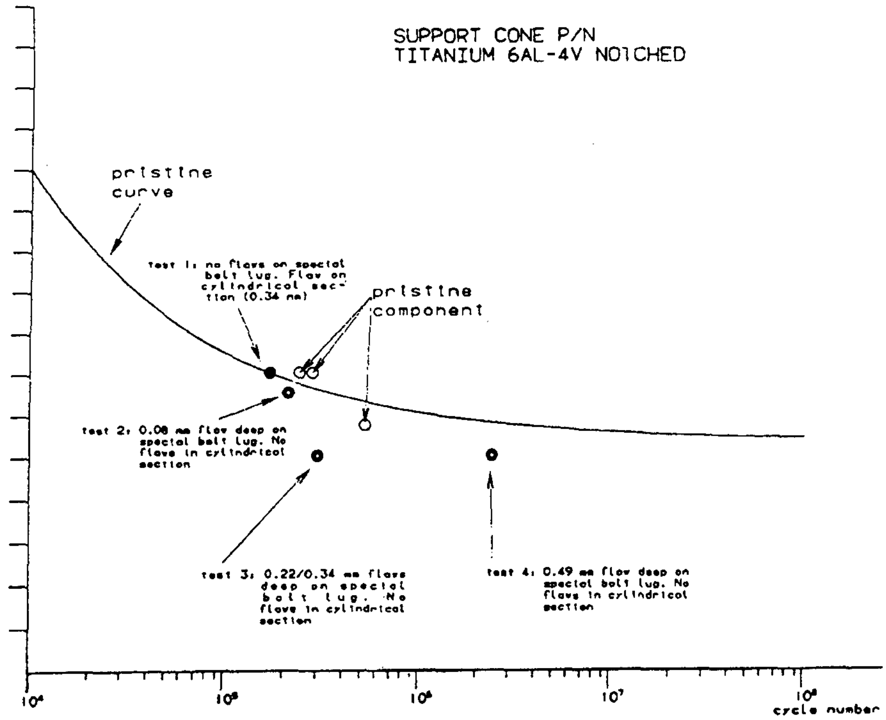

Figure 10 - M.R. Hub Support Cone Flaw Tolerance Data

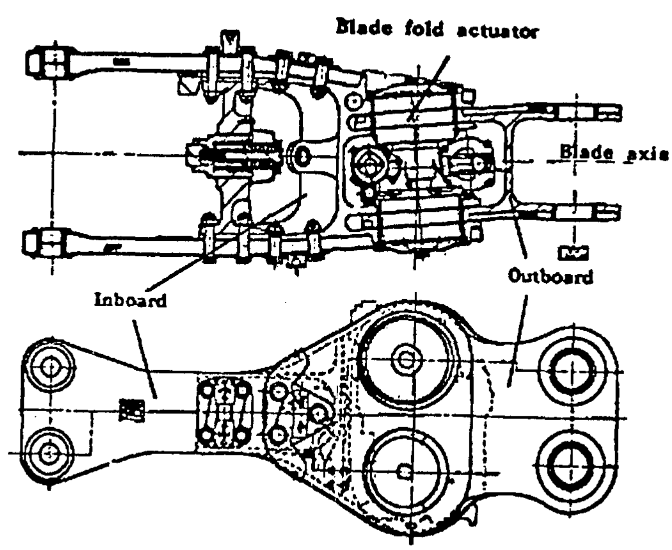

Figure 11 - M.R. Inboard and Outboard Tension Links

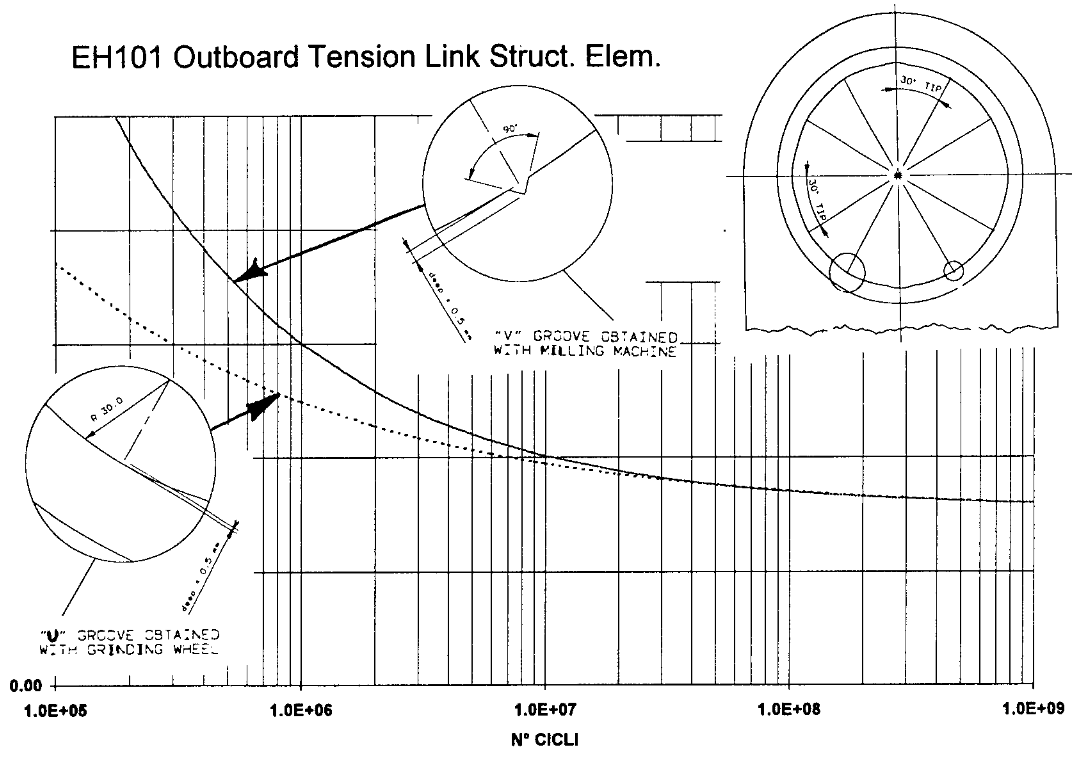

Figure 12 – Titanium Lug Elements - Flaw Tolerance Data

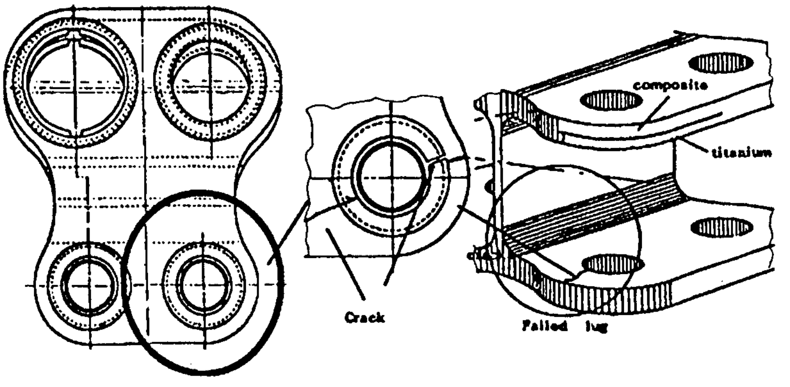

Figure 13 - M.R. Outboard Tension Link fatigue crack of the titanium lug

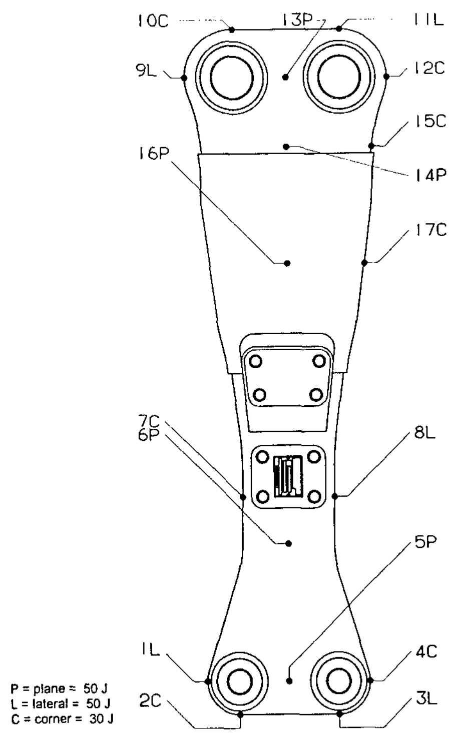

Figure 14 - M.R. Civil Tension Link Impact Damages for D.T. Test

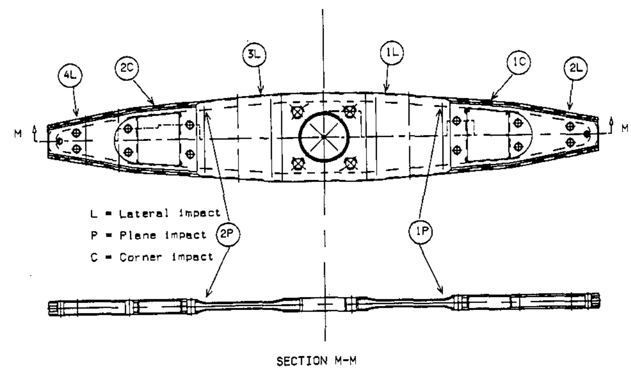

Figure 16 – Tail Rotor Hub Impact Damages for Test

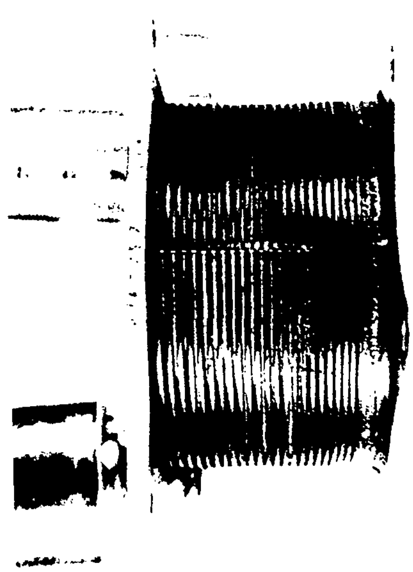

Figure 15 – Tail Rotor Elastomeric Bearing in Contamination Tests

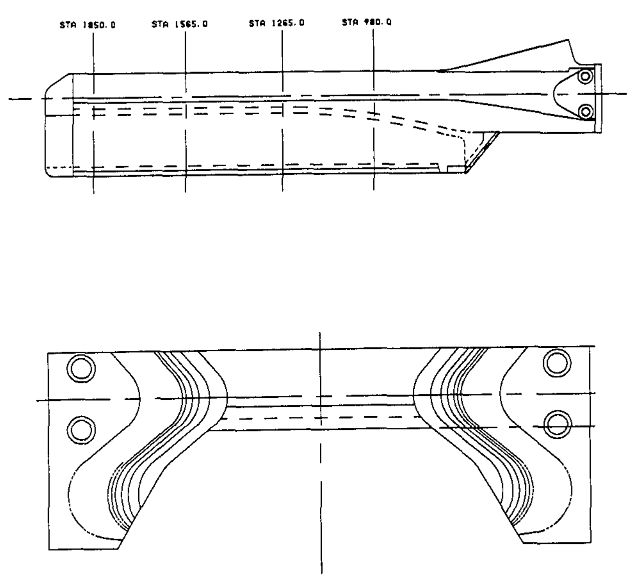

Figure 17 – Tail Rotor Blade D.T. of Trailing Edge

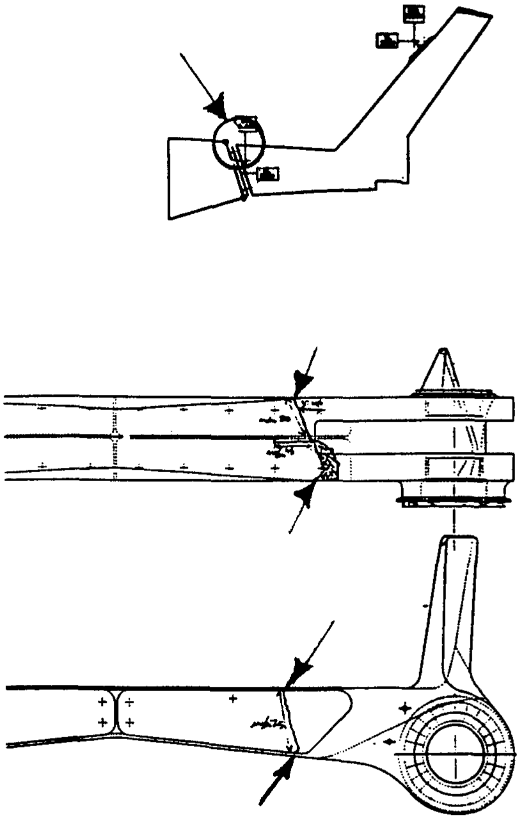

Figure 18 – Naval Rear Fuselage D.T. of Folding Beams

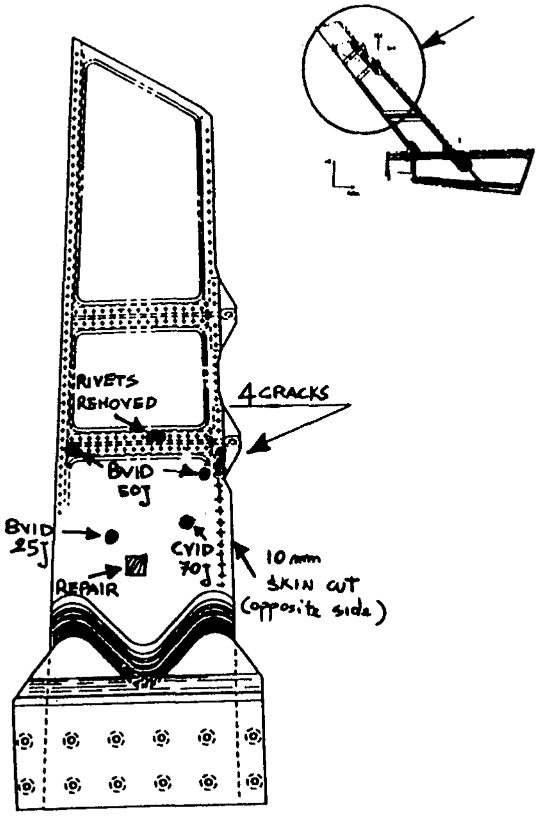

Figure 20 - EH101 Tail Unit subcomponent for flaw tolerance test - Fin Area and 90° Gearbox Attachments

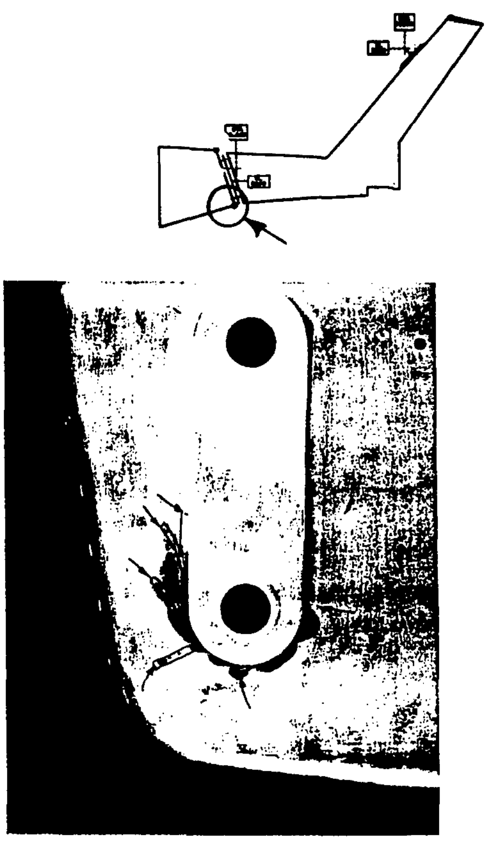

Figure 19 – Civil Rear Fuselage – End Fittings Slow Crack Growth

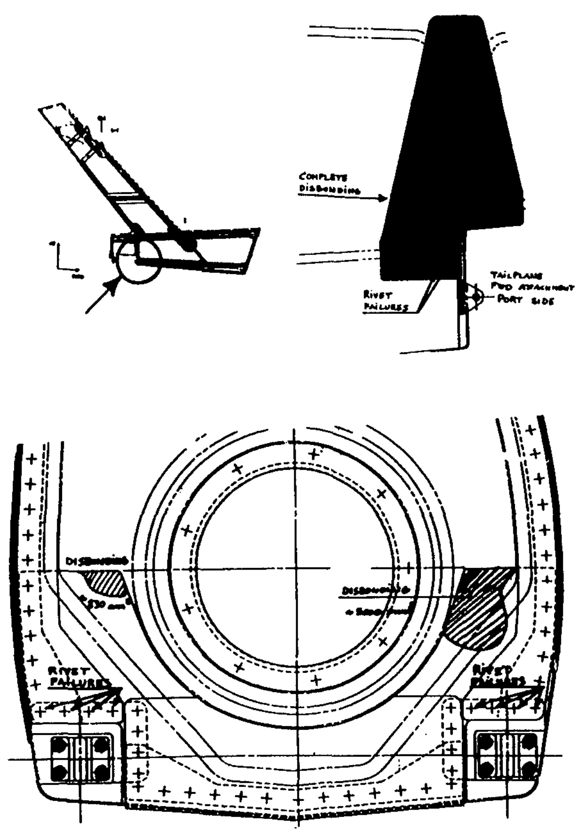

Figure 21 - EH101 Tail Unit subcomponent for flaw tolerance test - Tailplane FWD Attachments Area

Factor K Related To The Likelihood Of Accidental Damage And Visibility For Inspection		Likelihood of Accidental Damage		
		Probable	Possible	Unlikely
Visibility of SSI for Inspection	Poor	5	4.5	4
	Adequate	4.5	4	3.5
	Good	4	3.5	3
Factor H Related To The Sensitivity To Damage		Static Strength Margin Of Safety (MS)		
		$MS \leq 0.33$	$0.33 < MS \leq 1.00$	$MS > 1.00$
Safe Life (L)	$L \leq 2500$	4	3	2.5
	$2500 < L \leq 10000$	3.5	2.5	2
	$10000 < L \leq 40000$	2.5	2	1.5
	$40000 < L$	2	1.5	1

SSI Accidental Damage Analysis By The Factored Safe Life Method:

Inspection Interval = (Undamaged SSI Safe Life) / (H × K)

Table 1 - SSI Accidental Damage Analysis by the Factored Safe Life Method -

COMPONENT	N° OF TESTS FOR SAFE LIFE OF METAL COMPONENTS AND FLAW TOLERANCE OF COMPOSITE COMPONENTS	N° OF TESTS FOR DAMAGE TOLERANCE
MAIN ROTOR HUB	2 Full Scale 10 Structural Elements	1 Full Scale
M.R. DAMPER HUB ATTACHMENTS	5 Full Scale	2 Full Scale
M.R. SUPPORT CONE	4 Full Scale	4 Full Scale
M.R. INBOARD TENSION LINK	4 Full Scale 30 Structural Elements	2 Full Scale 10 Struct. Elements
M.R. OUTBOARD TENSION LINK	3 Full Scale 20 Structural Elements	3 Full Scale 12 Struct. Elements
M.R. CIVIL TENSION LINK	2 Full Scale	1 Full Scale
MAIN AND TAIL ROTOR ELASTOMERIC BEARINGS	9 Full Scale	2 Full Scale
TAIL ROTOR BLADE	6 Full Scale	2 Full Scale
TAIL ROTOR HUB	2 Full Scale 30 Structural Elements	2 Full Scale

Table 2 - Main and Tail Rotor Test Plan for Damage Tolerance -